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67982/001

2. Patent application number
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3. Full name, address and postcode of the or of each applicant *(underline all surnames)*

KEMP, Malcolm Hugh David
29 Woodwarde Road
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SE22 8UN

Patents ADP number *(if you know it)*

66257350001

If the applicant is a corporate body, give the country/state of its incorporation

4. Title of the invention

A Solar-Powered Orbital Launch Vehicle and Solar Sail Design

5. Name of your agent *(if you have one)*

"Address for service" in the United Kingdom to which all correspondence should be sent *(including the postcode)*

PAGE HARGRAVE
Southgate, Whitefriars
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Patents ADP number *(if you know it)*

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Country	Priority application number *(if you know it)*	Date of filing *(day / month / year)*
GB	0224221.2	17/10/2002

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Keith Gymer
020 7240 6933

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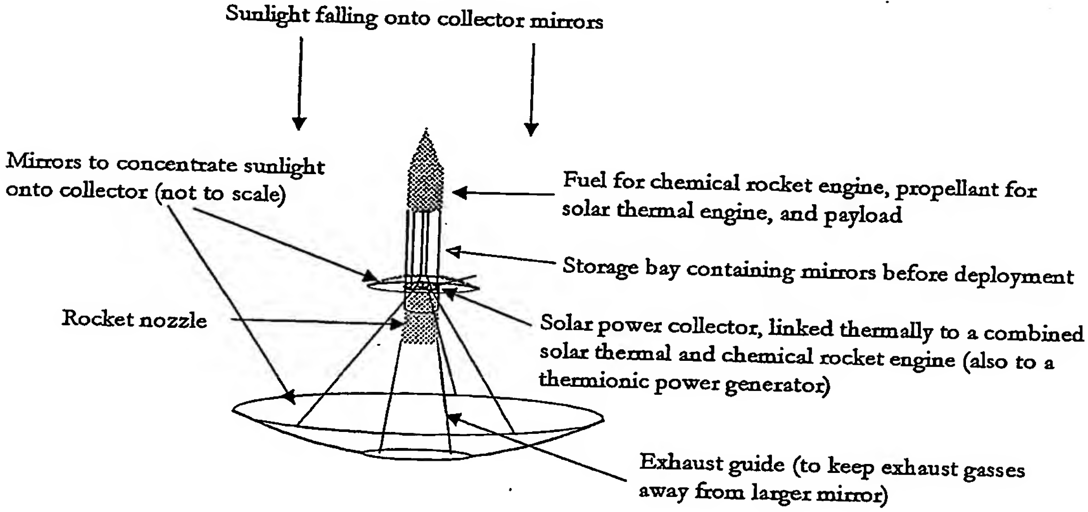
A Solar-Powered Orbital Launch Vehicle and Solar Sail Design

1 Introduction and Summary

1.1 All current orbital launch vehicles rely on chemical rockets. Once in orbit, some vehicles now use engines powered by electricity generated from sunlight to move from low earth orbit (LEO) to geostationary earth orbit (GEO) or further, or for attitude control when the desired orbit has been reached. Launch of an operational solar sail is also planned in the near future.

1.22 The purpose of this paper is to explore the possibility of using solar energy to power a vehicle **before it reaches orbit**, as well as for orbital transfers etc. The proposed approach is illustrated in Figure 1. A concentrator design is proposed that has several advantages over conventional parabolic arrangements. The concentrator design is described further in Appendices A and B.

Figure 1



1.3 There are several ways in which the dynamics of pre-orbital solar powered rocketry differ from those applicable to chemical rocketry. With chemical rocketry, it is in principle optimal to accelerate as fast as the payload allows, so that gravity has as little time as possible to slow the vehicle down. With solar-powered rocketry, this is no longer true, as the longer the vehicle is in flight the more energy it can collect. Appendix C considers these differences further and calculates the theoretical power requirement and hence minimum collector aperture size (per unit payload mass) that a solar powered launcher requires to reach earth orbit. If the vehicle starts at rest then the theoretical minimum aperture size is about 45 m^2 per kg.

A mirror arrangement as per Figure 1 can involve a total collector area that is only modestly larger than the effective aperture area. NASA has recently developed ultra-lightweight mirrors that weigh circa 5 gm^{-2} (in part for solar sail purposes). If the vehicle's net conversion efficiency of solar energy into thrust were c. 50 – 60% (which might be achievable using solar thermal engines) then such mirrors would in principle be light enough to permit such a launcher to lift itself *and* perhaps a roughly equal weight of payload into orbit.

1.4 In practice, some further improvements in the power to weight ratio would make such a solar powered orbital launch vehicle more practical, and it may in any case be better to consider a

vehicle that is powered by chemical rocketry early on in its flight, switching to solar power part of the way to orbit, to reduce air resistance early on in flight.

1.5 There are several ways in which concentrated sunlight can be converted into thrust. Appendix C shows that the closer such a launcher is to reaching orbital speed, the higher is the optimal specific impulse that its engine(s) should be delivering. Appendix D discusses the merits of different propulsion technologies that could be coupled with such concentrators. The proposed concentrators have the potentially important payload (and hence cost) advantage that they can be re-used with a range of propulsion technologies, each one suited to a different part of a spacecraft's flight:

Point in trajectory	Probable optimal propulsion technology
(a) Immediately after launch	Chemical rocketry (to limit air resistance), or possibly lifted by a balloon into the upper atmosphere (or possibly air breathing solar thermal propulsion).
(b) Approaching LEO	Solar thermal propulsion seems likely to be more efficient overall than using solar powered electric engines, such as ion engines.
(c) Between LEO and GEO	Various possibilities are available including solar thermal and solar-powered electric engines.
(d) At GEO	Solar power could be used for satellite operation and attitude control
(e) Beyond GEO	The faster and further away from the earth the spacecraft travels, the more attractive in principle become solar sails, except very far away from the sun (but see Appendix D).

A particularly interesting feature of the proposed mirror lay-out is that it can be reused with essentially no modification as a solar sail once the vehicle is sufficiently far from the earth's atmosphere. An important advantage it offers in this context is that it would offer a higher thrust per unit mirror area than traditional solar sail designs, which generally require the mirrors to be positioned at reasonably oblique angles to the incident sunlight. The vast majority of the mirror area in the proposed lay-out is nearly perpendicular to the sun's rays.

If it were possible to combine the ignition chamber of a chemical rocket with the main heating chamber of a solar thermal propulsion engine then further weight savings might be possible were the vehicle to use chemical rocketry immediately after launch, which might make creation of a totally reusable orbital launch vehicle more feasible.

1.6 The proposed concentrator layout might also offer advantages in non-astronautical contexts, including electric power generation for long duration high altitude balloon flights or in a terrestrial context, perhaps then contained within balloons that remain tethered to the ground.

2. The design of the proposed solar power collectors

2.1 I have a patent application (PCT/GB01/01161) that amongst other things claims to offer attractive ways of constructing very lightweight highly efficient solar power concentrators. The basic logic is:

(a) There is an upper limit imposed by the second law of thermodynamics on how much you can concentrate black body radiation such as sunlight. It is impossible to concentrate such radiation to a temperature greater than the temperature of the source of the radiation. For sunlight, this limit is the temperature of the sun's photosphere (circa 5900 K). The effective upper limit is a little lower, c. 5000 K, at ground level, because of atmospheric attenuation.

(b) It is possible to get very close to this thermodynamic upper limit with a relatively simple design involving two suitably positioned mirrors shaped in a rotationally symmetric fashion, as per Appendix A.

(c) If such mirrors are very thin then they can be kept very lightweight, perhaps so lightweight that they are not even rigid. Of course, they still need to be kept in the right shape, but even if they are not rigid this can be achieved either by spinning them (i.e. using centrifugal forces) or by embedding them in a lightweight inflated balloon structure.

(d) My patent application notes the possibility of converting the concentrated sunlight to thrust either via heating the propellant directly (i.e. via solar thermal propulsion) or by concentrating the sunlight onto a thermionic power generator (and then using the electricity thus generated to power an ion drive or other similar propulsion system).

2.2 The suggestions noted in 2.1(d) are similar to those proposed for the Solar Orbit Transfer Vehicle (SOTV), an experimental design developed recently by Boeing under a $48m contract from the US Air Force Research Laboratory, Kirtland Air Force Base, New Mexico. The SOTV aimed to use solar thermal propulsion to move a payload, **once in orbit**, from LEO to GEO at a leisurely pace (c. 15-30 days), delivering a low thrust but with a high specific impulse (750-800 seconds, versus the c. 450 delivered by a liquid hydrogen/liquid oxygen chemical rocket as used by, say, the NASA Space Shuttle). The SOTV envisaged concentrating sunlight using parabolic mirrors into a cavity in which hydrogen gas was heated to circa 2300 K. The resulting expansion of the hydrogen gas would provide the thrust. Hydrogen is the optimal propellant to use because of its low molecular mass, which all other things being equal raises the specific impulse that the engine delivers, minimising the propellant required and thus increasing the useful payload delivered to GEO. The SOTV also envisaged using the parabolic mirrors in conjunction with a thermionic power converter to provide power to the payload once at GEO.

2.3 However, the proposed mirror layout differs materially from the parabolic mirrors proposed for the SOTV. Advantages include:

(a) Parabolic mirrors can in principle only achieve concentrations that are circa one-quarter of the thermodynamic upper limit, see [1]. This reduces by about 30% the temperature to which it is possible to heat up the concentrator onto which the sunlight is concentrated, with a consequential reduction in thrust and in the efficiency of conversion of sunlight into propellant energy. It also increases the size and hence weight of the required concentrator. For example, the specific impulse arising from using hydrogen gas appears to rise from the circa 750 – 800 seconds available at an operating temperature of 2300 K to over 1000 seconds if the operating temperature is increased to 3700 K. This is potentially achievable with the proposed mirror design whilst still remaining under the melting point of graphite or tungsten.

(b) It seems likely to be desirable for the vehicle to be symmetric and travelling towards the sun. A parabolic mirror would probably need to have a hole in it to permit propellant to be fired out backwards, which would further reduce its overall efficiency.

(c) Several possible mirror layouts arise using the ideas contained in the above patent application depending on the initial seeding parameters chosen. Certain variants appear to have surface areas that are only very modestly larger than the aperture area onto which the

sunlight falls. These layouts should therefore also weigh less than a corresponding parabolic mirror arrangement.

3. Potential engineering challenges

3.1 An obvious challenge is that of making the mirrors sufficiently thin yet strong. This challenge is similar to that faced by "solar sails". Significant improvements have recently been made in this field, and further improvements may be possible, see Appendix D.

3.2 The mirrors would probably in part need to keep their rotational symmetry via centrifugal forces, i.e. by being spun (around their axis of rotation). Such spin could perhaps be imparted by air passing over suitable propeller blades or fins (e.g. as in Figure 1) as the arrangement passed through the (high upper) atmosphere.

3.3 A possible complication is the need to keep the mirrors from collapsing in on themselves as the launcher accelerates upwards. A solar powered launcher would in ideal conditions accelerate at 1g or less (see Appendix C), i.e. a significantly lower acceleration than with a typical chemical rocket launcher. As mentioned above there are several possible mirror layouts depending on the choice of parameters used to seed the iterative process set out in the above patent application. The one that seems to involve the least mirror weight per unit aperture and which involves the least likelihood of the engine exhaust fouling the sun's rays as those rays approach the solar thermal engine also has the advantage that the main mirror being used to collect the sunlight would be dragged along behind the solar thermal engine. This, in combination with rotation, would naturally facilitate retention by that mirror of its desired shape. The second mirror is much smaller than the first one (and closer to the engine) so a spur would probably be needed pointing forwards from the engine, which can tether the second mirror to keep it in the required shape in combination with it being rotated.

Alternatively, we might embed the mirrors into (or support the mirrors within) a balloon structure.

3.4 The launcher needs to point (quite accurately) towards the sun throughout its launch. If the mirror shape is being achieved or enhanced by rotation around its axis of symmetry then the mirrors will have some of the characteristics of a gyroscope. It should therefore be possible to maintain the desired direction for the vehicle, at least for a short period of time, although this might be more difficult if the vehicle were still within the atmosphere because of air resistance. The flight time is likely to be circa 10 – 30 minutes (see Appendix C), during which time it should be possible to keep the launcher's axis pointing towards the sun by modest adjustments to the direction of thrust. But there may be constraints on the time of day when it is most efficient to launch since it appears to be optimal for much of the flight to eject propellant at about 45° to the vertical.

4. Other comments

4.1 The proposed launch methodology if successful would most likely be significantly less expensive than existing chemical rocket based launch methodologies, since construction of large very lightweight mirror designs should be quite cheap if made in bulk. Because of the potential for multiple mode operation, it may be possible to make the launcher wholly reusable.

4.2 For re-entry, you might be able to slow down the vehicle by, in effect, using the mirrors as parachutes, taking advantage of air resistance. Or, the launch trajectory might be reversed, if you had available an additional supply of propellant. Or you might stow away the mirrors and re-enter the earth's atmosphere in a more conventional manner.

4.3 Perhaps worth noting in this context is that in a very rarified atmosphere, gas molecules can be expected to behave quasi-ballistically, and therefore would become concentrated at the image point (in the same way as light rays) if the mirror arrangement is pointed along the line of the trajectory into the direction of the air flow. This could be problematic if the converter of the solar power into more usable energy thus became damaged. Alternatively, it could be beneficial if the concentrated gases could be used for other purposes, e.g. deflected or propelled back out again to slow descent. However, the mirror layout highlighted above as most attractive for solar thermal propulsion may not be the one that is most ideal for this purpose.

In principle this effect could also supply propellant on the way into orbit. This might increase the effective power to weight ratio, making improvements in mirror weight less necessary before such a launcher could become practical, but this possibility is not considered further in this paper.

4.4 A more plausible way of avoiding the use of internally carried propellant whilst within the atmosphere would be to use an "air-breathing" solar thermal propulsion engine. Such an engine would be the solar thermal equivalent to a gasoline powered jet engine. An air intake would gather in air, and compress it, before it was heated by passage through or over the plate heated by the concentrated sunlight being generated by the mirror pair. The heated air would be ejected at high speed to provide the required thrust, As with a jet engine, some of the ejected gas could be made to pass through a propeller mechanism that drives the intake compressor.

Whilst an air-breathing solar thermal propulsion engine might be an interesting novelty toy, or a means of generating thrust for, say, a high altitude balloon, the difficulties that others seem to have faced when attempting to develop practical air breathing space planes suggests that it would be challenging to achieve significant acceleration from such an engine.

But, if such an engine could be integrated with a solar thermal rocket engine then this would reduce the weight penalty it would otherwise incur. As long as the thrust was sufficient to overcome gravity, it could lift the vehicle slowly in the lower reaches of the atmosphere, to minimise air resistance arising from the large mirror, only picking up speed in the upper reaches (where in any case greater speed might be desirable to collect sufficient air to generate the required thrust).

4.5 The technology proposed in this note might also provide:

(a) A cost effective source of power on the ground, perhaps by embedding the mirrors in a balloon structure that was lighter than air but kept tethered to the ground. The possibility of creating a power station that can generate many kilowatts of power per 1 kg of weight could offer some attractive commercial opportunities unless fabrication costs are very high or reliability/lifetime is poor.

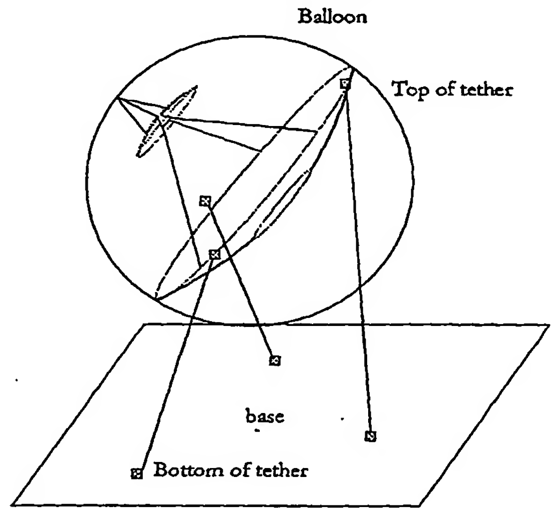
(b) A useful source of significant amounts of power to very long duration balloon flights. NASA has already tested super-pressure balloons capable of staying in the upper atmosphere for many days. Some of the engineering difficulties highlighted above in the context of a space launcher (such as sensitivity to air resistance, need for ultra-lightweight mirrors) become less important in an atmospheric balloon context.

(c) A method of maintaining hot air balloons (as opposed to super-pressure balloons) in the (high) atmosphere almost indefinitely. In the presence of direct sunlight the air outside the balloon will expand as the external pressure rises. However, with a sufficiently efficient converter of sunlight into power and a sufficiently efficient heat pump, you should always be able to heat the air inside the balloon more rapidly than the air outside is being heated, thereby always retaining uplift. At night you might need to supplement the heating using energy stored in a battery.

Any such solar concentrator would require a tracking system to keep it pointed towards the sun. A cost effective way of doing this might be to tether the balloon using several (at least three) different ropes or wires joined to different parts of the balloon and to different anchoring points.

For example, suppose for (a) that the mirrors were in a spherical balloon filled with a light gas such as hydrogen or helium. Suppose counterweights are if necessary introduced so that the centre of gravity of the combination of balloon, mirrors, power generator and counterweights is roughly at the centre of the balloon, but with the combination still lighter than air. Suppose also that three tethers were joined to this balloon at points spaced equally around the great circle coaxial with the mirrors, as in Figure 2. If necessary the entire arrangement could be housed within a transparent cover, to avoid the balloon being blown about by the wind.

Figure 2



Then just by changing the lengths of the individual ropes (e.g. wrapping or unwrapping them around a spindle), it would appear to be possible to get the mirrors to point in any desired direction likely to be relevant to a solar power concentrator. This is possible because of the interaction between the uplift being provided by the inflated balloon and the tension provided by the tethers. The positioning could be changed over time according to some preset timing mechanism (that varied by date), and/or via some feedback mechanism linked to the power being generated (and/or some other sensing mechanism). The power generated by the concentrator could be carried away from the balloon via wire tethers or via separate power leads.

An equivalent approach can be used for (b) and (c) with the balloon no longer being tethered to the ground, but with the weight of the payload providing the equivalent anchoring. However, some additional mechanism for keeping the balloon from rotating horizontally would also be needed, e.g. a small propeller engine.

References

1. R. Winston, "Nonimaging optics". Scientific American, March 1991.

2. The US National Research Council Committee on Thermionic Research and Technology "Thermionics Quo Vadis? An Assessment of the DTRA's Advanced Thermionics Research and Development Program", National Academy Press (2001).

APPENDIX A

Positioning of a two-mirror arrangement capable of focusing sunlight to almost the thermodynamic upper limit

A.1 The layout is rotationally symmetric, and therefore can be determined by considering a cross-section through this axis of symmetry (say in the *xy*-plane, with the *x*-axis being the axis of symmetry).

A.2 We iteratively determine the positioning of each consecutive point on each of the two mirrors as follows:

(a) At the t 'th iteration, the mirrors are positioned so that a light ray starting at exactly the object point $(f, 0)$ will go through the image point (0, 0), after striking the first mirror at $M_1(t) = (m_{1,x}(t), m_{1,y}(t))$ and the second mirror at $M_2(t) = (m_{2,x}(t), m_{2,y}(t))$. The angle that the two mirrors make to the *x*-axis at the t 'th iteration can be found by geometry, given behaviour of light rays when they are reflected off of surfaces. These angles then define the tangents to the two mirrors at this step in the iteration.

(b) The position of $M_2(t+1)$ is found so that if a light ray starts at (f, h), strikes the first mirror at $M_1(t)$ and the second mirror at $M_2(t+1)$ then it passes through $(0, ZBh)$, for a suitable constant B (the magnification of the arrangement), and a suitable sign $Z = \pm 1$, h small (positive or negative). We apply the additional constraint that $M_2(t+1)$ must be on the tangent to the second mirror as identified in (a), which means that the (unique) position of $M_2(t+1)$ can be found by geometry.

(c) The position of $M_1(t+1)$ is found so that if a light ray starts at $(f, 0)$, strikes the first mirror at $M_1(t+1)$ and the second mirror at $M_2(t+1)$ then it passes through (0, 0). Again we apply the additional constraint that $M_1(t+1)$ must be on the tangent to the first mirror as identified in (a), which means that the (unique) position of $M_1(t+1)$ can also be found by geometry.

(d) The angles that the mirrors make to the *x*-axis at the $(t+1)$'th iteration and hence their tangents are then up-dated as per (a), and the iteration repeated.

A.3 In the limit as $h \to 0$ this approach defines the shape of two surfaces that together form a perfectly aplanatic (of order 1) imaging mirror arrangement.

A.4 The mathematics can be restated as follows:

(a) $M_0(t) \equiv (m_{0,x}(t), m_{0,y}(t))$ and $M_3(t) \equiv (m_{3,x}(t), m_{3,y}(t))$ define the positions of the centres of the object and image planes respectively (so $M_0(t) \equiv (f,0)$, $M_3(t) \equiv (0,0)$ say)

(b) $d_i(t)$ (for $i = 0, \ldots, 2$) is the angle that a ray from $M_i(t)$ to $M_{i+1}(t)$ makes to the *x*-axis

$$d_i(t) = \arctan\left(\frac{m_{i+1,y}(t) - m_{i,y}(t)}{m_{i+1,x}(t) - m_{i,x}(t)}\right)$$

(c) $p_i(t)$ (for $i = 0, \ldots, 2$) is the distance between $M_i(t)$ and $M_{i+1}(t)$, i.e.

$$p_i(t) = \sqrt{\left(m_{i+1,x}(t) - m_{i,x}(t)\right)^2 + \left(m_{i+1,y}(t) - m_{i,y}(t)\right)^2}$$

(d) $a_i(t)$ is the angle that a tangent to the ith surface makes to the x-axis, taking $a_0(t)$ and $a_3(t)$ to be the angles that the object and image planes make to the x-axis (being 90° for all t for the surfaces to be rotationally symmetric about the x-axis), i.e. for reflection (for $i = 1$ and 2):

$$a_i(t) = \frac{d_i(t)}{2} + \frac{d_{i-1}(t)}{2}$$

(e) The values of $m_{i,x}(0)$ and $m_{i,y}(0)$ (for $i = 1$ and 2) are chosen so that the resulting mirror layout satisfies the sine criterion. For a far away source (say along the negative x-axis, say $f = -10^9$) we can without loss of generality set b, a scale parameter, equal to unity, setting $B = b/p_0(0) = 1/p_0(0)$. The sine criterion is then satisfied if:

$$m_{1,y}(0) = \pm \frac{m_{2,y}(0)}{\left(m_{2,x}(0)^2 + m_{2,y}(0)^2\right)^{1/2}}$$

(f) We can iterate either from shallower angles to more oblique angles, or vice versa. We can switch between the two by using as seed values later iterated results and reversing the sign of h. If we iterate from highly oblique angles then we can choose $m_{2,x}(0)$ equal to a small number close to zero, say 10^{-9}, and we can choose $m_{1,x}(0) = q_1$ and $m_{2,y}(0) = q_2$, where q_1 and q_2 are arbitrary real numbers (positive or negative). Then $m_{2,y}(0)$ would need to be ±1 for the initial parameters to satisfy the sine criterion. Without loss of generality we can choose $m_{2,y}(0)$ to be -1.

(g) The choice of the sign Z is made so that the sine criterion remains satisfied as t changes. Only one choice will work depending on the layout, for the two-mirror layout being analysed here, and if $m_{2,y}(0) = -1$, then it seems that the correct choice is $Z = \mathrm{sgn}(m_{2,y}(0))$.

(h) We update the values of $M_i(t)$ as follows for a small value of h (which can also be positive or negative):

$$M_i(t+1) \equiv \begin{pmatrix} m_{i,x}(t+1) \\ m_{i,y}(t+1) \end{pmatrix} = M_i(t) + w_i \begin{pmatrix} \cos(a_i(t)) \\ \sin(a_i(t)) \end{pmatrix} h$$

where $r_{i-1}(t) = \sin\left(a_{i-1}(t) - d_{i-1}(t)\right)$ $s_i(t) = \sin\left(a_i(t) - d_{i-1}(t)\right)$ and where

$$w_2 = \frac{p_1 r_0}{p_0 s_2} \quad w_1 = -ZB\frac{p_1 s_3}{p_2 r_1}$$

(i) We end the iteration no later than when light rays cease to be able to pass freely through the mirror arrangement, once it has been rotated as in (j).

(j) The complete, three-dimensional mirror surfaces can then be found by rotating the curves produced above around the x-axis.

A.5 Each of the signs of q_1, q_2 and h can be either positive or negative leading to 8 possible two mirror layouts. The visual appearance and characteristics of the layout also depends on whether abs($m_{2,y}(0)$) is larger or smaller than unity. For the purposes considered in this note, it is unlikely to be desirable for abs($m_{2,y}(0)$) to be larger than unity, since it would significantly increase the overall surface area of the mirrors, and hence the weight of the solar power collector.

A.6 Examples of cross-sections that arise with different choices of signs of q_1, q_2 and b and size of abs($m_{2,y}(0)$) are shown below in Figure 3. These examples concentrate on $q_1 = \pm 2$, $q_2 = \pm 0.2$ (or ± 3 when considering cases where for abs($m_{2,y}(0)$) is larger than b, i.e. unity):

Figure 3 – cases where abs($m_{2,y}(0)$) is smaller than unity

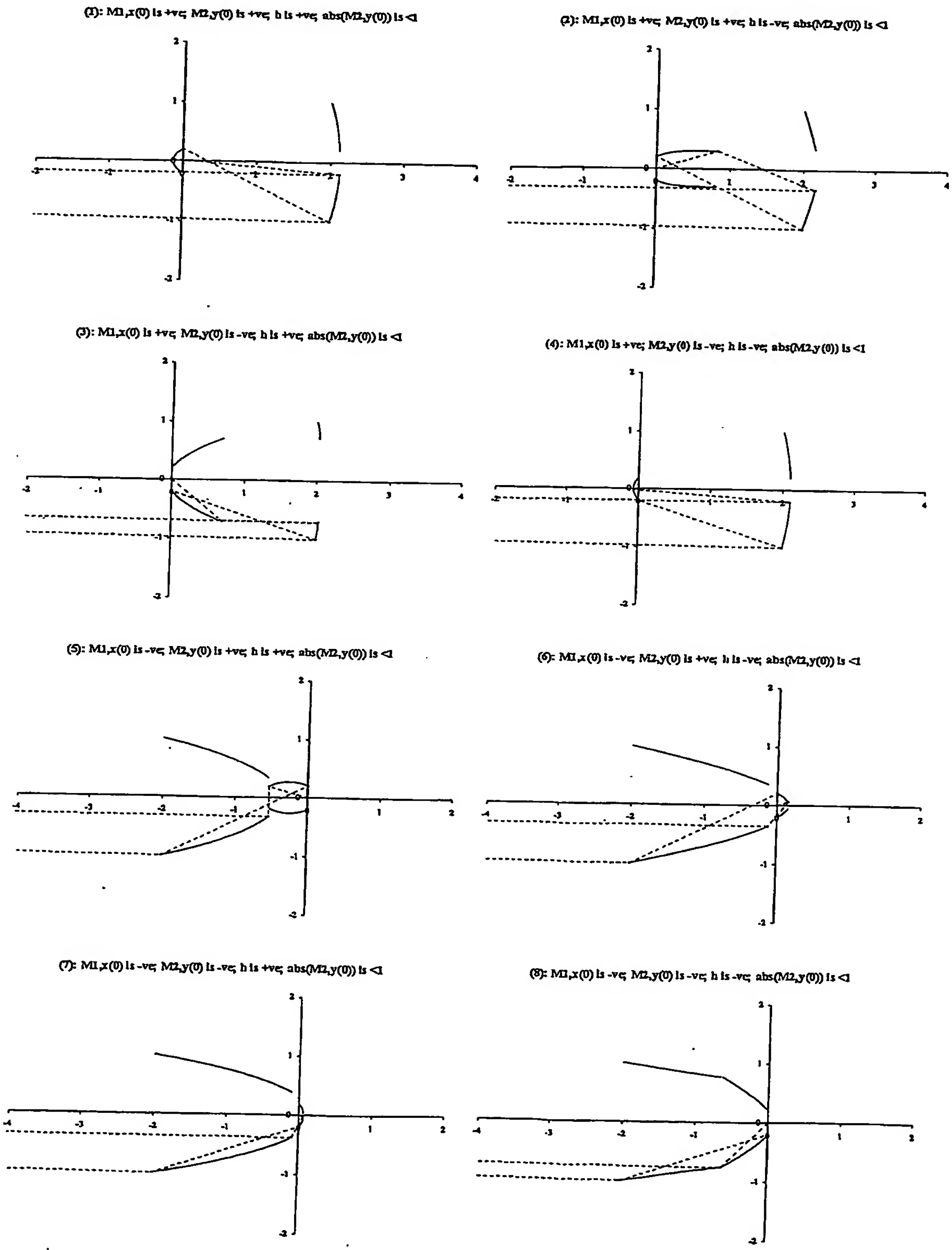
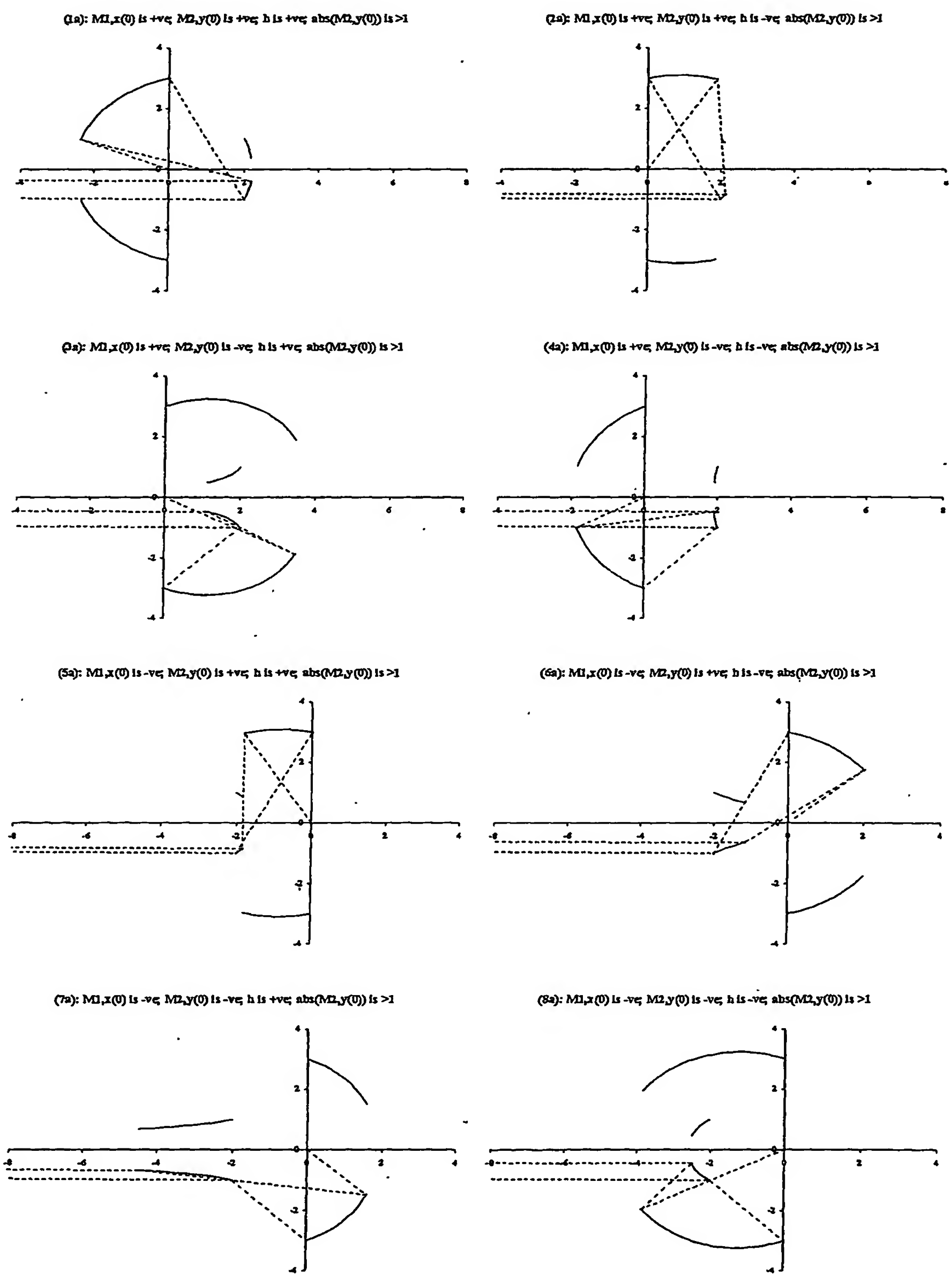


Figure 3a – cases where abs($m_{2,y}(0)$) is larger than unity



In each of the above plots, the solid lines are cross-sections of the mirrors themselves, and the dotted lines are the paths of light rays from the object to the image passing through extremities of the mirror arrangement. One extremity is the ultra-highly oblique ray that effectively defines the start of the iteration process referred to in A.4. The other is as per A.4(i). Clearly, it would be possible to limit the iteration to a smaller range of angle spans by starting the iteration later or finishing it sooner than implied in the above plots.

A.7 In passing one can note that some special or degenerate cases exist to the iteration process, but they seem to have limited relevance to the purposes considered in this paper. Degenerate cases include cases where the second mirror is extremely small (achieved in the iterative process by setting $m_{2,y}(0)$ to a very small number). The first (and larger) mirror then takes the shape of a paraboloid, if the source is far away, or an ellipsoid if the source is nearby. When the first and second mirrors are the same size (and the magnification, B, is unity), then the mirror arrangement corresponding to plot (5) becomes a double confocal, co-linear, equally sized ellipsoidal arrangement.

A.8 The mirror layout which is given the most attention in the patent application PCT/GB01/01161 is that shown in plot (5), including the case where iteration is carried out from shallower to more oblique angles of incidence onto the image plane and where the seed parameters are chosen so that the two mirrors join up into a single surface. For a far away light source this can be achieved using the following initial conditions:

$$m_{2,y}(0) = -m_{1,y}(0) \Rightarrow \left(m_{2,x}(0)^2 + m_{2,y}(0)^2\right)^{1/2} = 1$$

One reason for this is that much of patent application PCT/GB01/01161 concentrates on high resolution optics, and having both mirrors "in front of" the image plane (in much the same way as a traditional microscope or photolithographic device operates) seems likely to be desirable. Such variants also normally have a relatively high coverage of possible angle spans, and a relatively low second order aberration factor.

The aberration factor (which is explained in the patent application) assesses the extent to which, say, light from the rim of the sun's outline in the sky would not fall exactly at the thermodynamic optimum would require (i.e. where it would fall if the mirror arrangement were aplanatic to higher than 1st order). For an ultra-high resolution imaging device, it is important that image aberration is kept as low as possible. For solar energy purposes, if the aberration factor is material, it is possible to improve the concentration characteristics of the mirror layout either by adjusting modestly the iteration process (so that circular objects fall more precisely on the same circle throughout the iteration process) or by adding more surfaces at which the light is deflected as it passes from object to image.

A.9 But for the purposes being discussed in this paper, plot (5) is undesirable because it has a relatively high ratio of actual mirror area to effective collector aperture perpendicular to the sun's rays, increasing the weight of the collector per unit power collected. The characteristics of the different plots in this respect are set out below:

Plot	$m_{1,x}(0)$	$m_{2,y}(0)$	**sgn(B)**	**abs($m_{2,y}(0)$)**	**effective aperture area**	**aberration factor**	**mirror surface area**
(1)	2	0.2	1	<1	96%	0.079	1.06
(2)	2	0.2	-1	<1	89%	0.017	1.56
(3)	2	-0.2	1	<1	50%	0.021	2.74
(4)	2	-0.2	-1	<1	96%	0.059	1.03
(5)	-2	0.2	1	<1	90%	0.020	2.86
(6)	-2	0.2	-1	<1	88%	0.038	3.24

(7)	-2	-0.2	1	<1	85%	0.077	3.51
(8)	-2	-0.2	-1	<1	41%	0.013	7.81
(1a)	2	3	1	>1	85%	0.008	17.46
(2a)	2	3	-1	>1	29%	0.009	41.80
(3a)	2	-3	1	>1	77%	0.007	31.57
(4a)	2	-3	-1	>1	77%	0.005	16.39
(5a)	-2	3	1	>1	27%	0.007	43.01
(6a)	-2	3	-1	>1	57%	0.012	23.46
(7a)	-2	-3	1	>1	54%	0.035	26.91
(8a)	-2	-3	-1	>1	80%	0.002	32.71

In the above table the items on the right hand side are calculated as follows:

effective aperture area = area of first mirror perpendicular to sun's rays, expressed as a proportion of the maximum possible were the angle span of rays falling onto the image plane to be the complete range from wholly oblique to exactly perpendicular to the image plane. The higher this is, the closer to the thermodynamic upper temperature limit such a concentrator can approach.

aberration factor = average maximum second order degree of aberration for sunlight in the vicinity of the earth (i.e. for a far away source subtending approximately a semi angle of 0.267°). The degree of aberration is calculated using the formulae contained in PCT/GB01/01161. Incidentally these formulae seem, by comparison with an explicit ray trace analysis, to provide a good approximation to the actual "true" aberration experienced for such a semi-angle, except at the extremes of the iteration, suggesting that most of the time higher order aberrations can be ignored. The lower the aberration factor, the closer the mirror pair can reach to the thermodynamic ideal (without resorting to adjustments of the sort referred to in A.8 or PCT/GB01/01161).

mirror surface area = the total surface area of the two mirrors combined as a multiple of the effective aperture area. The smaller is this figure, the less is the mirror surface area required per unit of power delivered, and therefore all other things being equal, the less the mirror arrangement will weigh.

A.10 The two lightest mirror layouts shown, and therefore the ones most likely to be helpful for the purposes considered in this paper, are those in plots (1) and (4). However, layouts akin to Plot (1) would have the disadvantage that the sunlight would pass very close to the positive x-axis on its way towards the image. Any exhaust gases propelled out of a solar thermal engine used in conjunction with such a mirror arrangement are likely to be ejected along approximately this axis. Indeed, it may be best to force them to be ejected along this axis, by some sort of exhaust guide, perhaps again made out of a very lightweight flexible material akin to the substrate underlying the mirrors. The light rays would then be fouled by the exhaust gasses themselves or this exhaust guide.

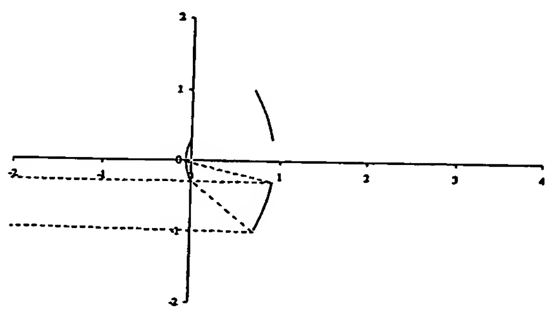
Better therefore are likely to be layouts akin to plot (4). The rays in this variant do not cross the positive x-axis. It also seems to offer a slightly lower surface area as a proportion of effective collector aperture, and better aberration characteristics without a altering the effective aperture area.

A.11 Changing the values of of q_1, q_2 but not their signs (or whether the absolute value of q_2 is smaller or larger than unity) produces mirror layouts that have visual similarities to the one shown in Plot 4 but are somewhat different shapes. Possible disadvantages of using values of $q_1 = 2$ and $q_2 = -0.2$ are that the larger mirror is some way away from the image. For a solar-powered orbital launcher, which presumably could be hundreds of metres in diameter, the further away the mirror is, the longer would need to be the wire or wires joining the engine to the mirror. This might therefore increase the masses of these wires. A more compact arrangement may be preferred. If the vehicle is to travel in the direction of its axis then

the solar thermal rocket exhaust needs to escape through the hole in the centre of the larger mirror, which subtends quite a small angle in this arrangement. This angle would be larger if abs(q_2) were larger and q_1 were smaller.

Of course such parameter changes also alter other characteristics of the mirror pair, in some instances detrimentally, so some trade-off is likely. For example, if Plot 4 is altered to use $q_1 = 0.7$ and $q_2 = -0.3$ then the mirror layout becomes as shown in Figure 3b. The average aberration factor improves to 0.020, but the effective aperture area falls to 91% and the mirror surface area as a multiple of the effective aperture area rises to 1.13. Reducing q_1 also makes the smaller mirror closer to the image plane, increasing the possibility that it might overheat.

Figure 3b



A.12 A simple way of implementing the above algorithm using Visual Basic code is set out in Appendix B, where the parameter $h0$ defines the speed of drawing the relevant mirror positions.

APPENDIX B

Sample Visual Basic code that can be used to identify mirror positioning as per Appendix A

```
Const tlim = 10000                    'say
Const pi = 3.14159265
Dim Mx(0 To 3, 0 To tlim) As Double, My(0 To 3, 0 To tlim) As Double
Dim d(0 To 3, 0 To tlim) As Double
Dim p(0 To 3, 0 To tlim) As Double
Dim a(0 To 3, 0 To tlim) As Double
Dim w(1 To 2, 0 To tlim) As Double

Sub identify_mirror_positioning(sht, Mx1init, My1init, Mx2init, My2init, Z, h0)
        Dim i As Integer, t As Integer, t1 As Integer, h As Double, B As Double
        'first establish the initial conditions
        For t = 0 To tlim
                Mx(0, t) = -10 ^ 9          'say (far away along negative x-axis)
                My(0, t) = 0
                Mx(3, t) = 0
                My(3, t) = 0
                a(0, t) = pi / 2
                a(3, t) = pi / 2
        Next t
        Mx(1, 0) = Mx1init
        My(1, 0) = My1init
        Mx(2,0) = Mx2init
        My(2,0) = My2init
        p(0, 0) = ((Mx(0, 0) - Mx(1, 0)) ^ 2 + (My(0, 0) - My(1, 0)) ^ 2) ^ 0.5
        h = (h0 / tlim) * p(0, 0)
        B = 1 / p(0, 0)
        'now carry out iteration, looping through to tlim
        For t = 0 To tlim - 1
                For i = 0 To 2
                        d(i, t) = Atn((My(i + 1, t) - My(i, t)) / (Mx(i + 1, t) - Mx(i, t)))
                        p(i, t) = ((Mx(i, t) - Mx(i + 1, t)) ^ 2 + (My(i, t) - My(i + 1, t)) ^ 2) ^ 0.5
                Next i
                For i = 1 To 2
                        a(i, t) = d(i, t) / 2 + d(i - 1, t) / 2
                Next i
                w(1, t) = -Z * B * (p(1, t) / p(2, t)) * Sin(a(3, t) - d(2, t)) / Sin(a(1, t) - d(1, t))
                w(2, t) = (p(1, t) / p(0, t)) * Sin(a(0, t) - d(0, t)) / Sin(a(2, t) - d(1, t))
                For i = 1 To 2
                        Mx(i, t + 1) = Mx(i, t) + w(i, t) * Cos(a(i, t)) * h
                        My(i, t + 1) = My(i, t) + w(i, t) * Sin(a(i, t)) * h
                Next i
        Next t
        'and print out positioning of mirrors, but perhaps only every 50'th iteration step
        For t = 0 To tlim – 1 step tlim \ 50
                t1 = t \ (tlim \ 50) + 2
                Worksheets(sht).Cells(t1, 1) = t
                Worksheets(sht).Cells(t1, 2) = Mx(1, t)
                Worksheets(sht).Cells(t1, 3) = My(1, t)
                Worksheets(sht).Cells(t1, 4) = Mx(2, t)
                Worksheets(sht).Cells(t1, 5) = My(2, t)

End Sub
```

APPENDIX C

The Continuous Constant Power Needed to Reach Orbital Velocity

C.1 With a conventional rocket launcher, the power that propels the fuel away from the rocket engine is provided by combustion of the propellant. Therefore, the greater the amount of propellant that can be ejected per unit time, the more rapid will be the acceleration of the rocket. Pre-orbit, this means that, all other things being equal, you should use the fuel up as rapidly as possible, to minimise the adverse impact that gravity has on the vehicle.

The dynamics of a solar powered launch vehicle are significantly different. The vehicle still needs propellant in order to impart sufficient momentum to overcome gravity, but there is less incentive to use the propellant up as rapidly as possible, as to do so will reduce the overall energy collected from the sunlight and therefore made available during the launch. It is instructive to work out what is the most efficient way of using the propellant that such a vehicle will need to carry.

C.2 Suppose that a launcher plus propellant has mass $m(t)$ and velocity $v(t)$ at time t (measuring t in seconds from launch), m' is the rate of change of m with respect to t, etc. Suppose that we have energy $E.dt$ arriving between t and $t+dt$ (E assumed to be constant). Take g to be the acceleration due to gravity (assumed constant at 10ms^{-2}). Assume orbital speed is 7700ms^{-1}. We ignore air resistance (i.e. we assume that any solar powered stage starts sufficiently high up).

C.3 Consider first the situation where we launch vertically and continue to travel vertically upwards, ejecting the propellant vertically downwards at a speed u (constant) relative to the launcher.

The launcher then satisfies the following equations of motion, derived from a consideration of conservation of energy and momentum:

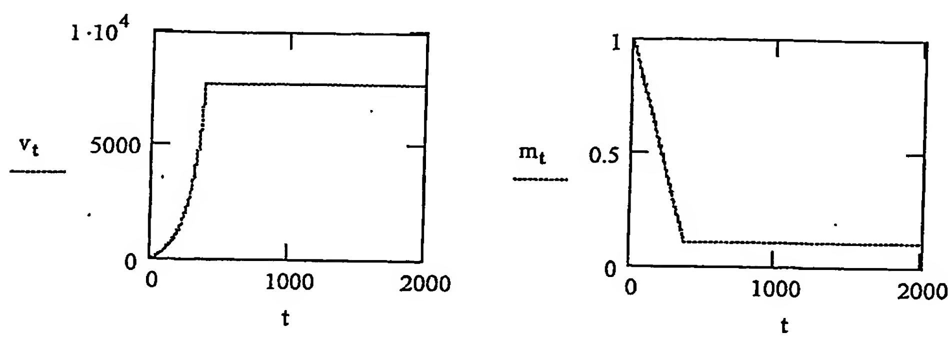
$$E = \frac{1}{2}mu^2$$

$$m(g + v') = -um'$$

We then have:

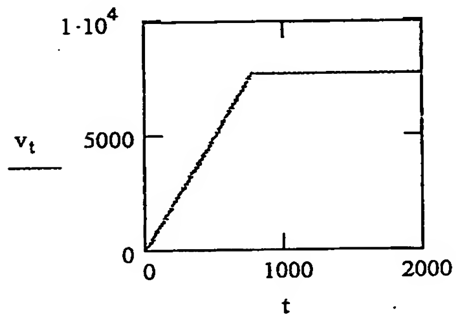
$$m' = -2\frac{E}{u^2} \qquad v' = -\frac{mg + um'}{m}$$

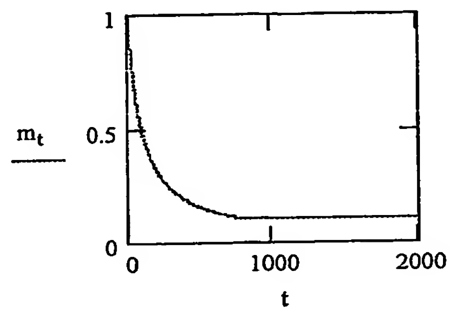
If $m(0) = 1$, $u = 5{,}000$ ms^{-1}, $E = 30{,}000$ then v and m progress as follows:



The acceleration starts low, but reaches c. 6.5g just before reaching orbital speed. The rate at which mass is ejected is constant. The ratio of initial to final mass is c. 10, so the power per unit final mass is 300 kW kg^{-1}.

C.4 Better is to allow u to vary. The optimal choice of u is to maximise v'/m' which occurs when $u = E/mg$. If $m(0) = 1$, $E = 8{,}000$ then v and m now progress as follows:





The acceleration is constant (at 1g) prior to reaching orbital speed. The ratio of initial to final mass is c. 20, so the power per unit final mass is c. 160 kW kg^{-1}. Mass is initially ejected at 800 ms^{-1}, but just before orbital velocity is reached, it is being ejected at c. 16,000 ms^{-1}.

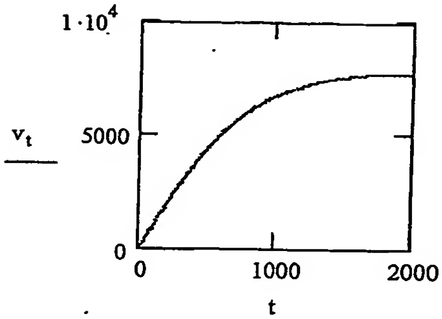
C.5 But you would not actually go vertically upwards to go into orbit. Instead, you need to end up going horizontally (if you want to end up in a circular orbit). The faster the horizontal speed, the greater the centrifugal effects offsetting gravity. If we assume that the optimal approach is to travel horizontally throughout the launch, then you would eject propellant out at an angle designed to just counteract the effect of gravity. If v is the horizontal velocity of the launcher, $vorb$ the orbital velocity, u the speed of ejection horizontally and w the speed of ejection vertically, then the equations of motion become:

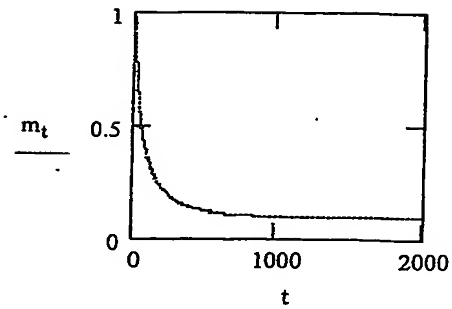
$$E = \frac{1}{2} m\left(u^2 + w^2\right)$$
$$m.v' = -u.m'$$
$$m.g\left(1 - \frac{v^2}{vorb^2}\right) = -w.m'$$

Again, you would choose u to maximise v'/m', which means that:

$$m' = -g^2 m^2 \frac{\left(vorb^4 - 2vorb^2 v^2 + v^4\right)}{E.vorb^4}$$
$$u = -\frac{\left(-2E.vorb^4 m' - m^2 g^2 vorb^4 + 2m^2 g^2 vorb^2 v^2 - m^2 g^2 v^4\right)^{1/2}}{vorb^2 .m'}$$

If $m(0) = 1$ and $E = 10{,}000$ then v and m now progress as follows:





The horizontal acceleration starts off at 1g but slows down as the effective pull downwards tails off as centrifugal effects bite. The propellant is initially ejected at a horizontal speed of 1,000 ms^{-1} and at a vertical speed of 1,000 ms^{-1}, i.e. with a total speed of c. 1,414 ms^{-1} at 45° to the vertical. Indeed it seems to continue to be optimal to eject propellant at this angle throughout the flight. The optimal propellant ejection speed rises substantially as the launcher reaches close to orbital velocity. The ratio of initial to final mass is c. 6, and so the power per unit final mass is now merely c. 60 kW kg^{-1}.

In practice, some upward movement would be necessary, to reduce air resistance (unless you started above the atmosphere), so the required power per unit mass would probably be a little higher than this.

C.6 In the analysis in C.5 the propellant ejection velocity rises without limit as the vehicle approaches orbital speed. It is also instructive to consider the optimal approach were there to be an upper limit on the propellant ejection speed.

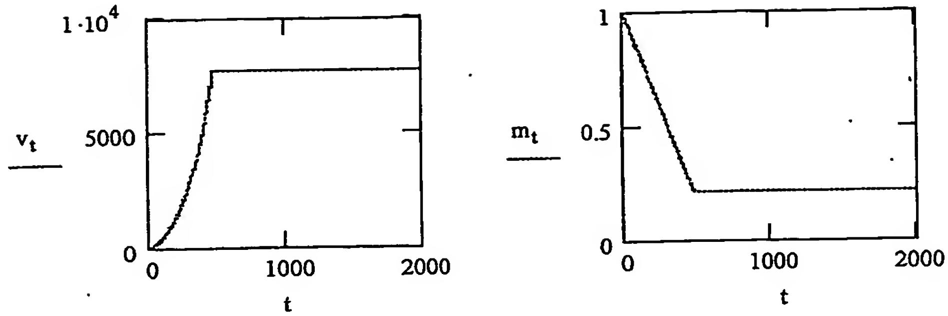
C.7 Consider first the situation where the propellant ejection speed is fixed at say p = 6,000 ms^{-1}. The equations of motion become:

$$E = \frac{1}{2}m\left(u^2 + w^2\right)$$
$$p^2 = u^2 + w^2$$
$$m.v' = -u.m'$$
$$m.g.\left(1 - \frac{v^2}{vorb^2}\right) = -w.m'$$

Again, you would choose u to maximise v'/m', which means that:

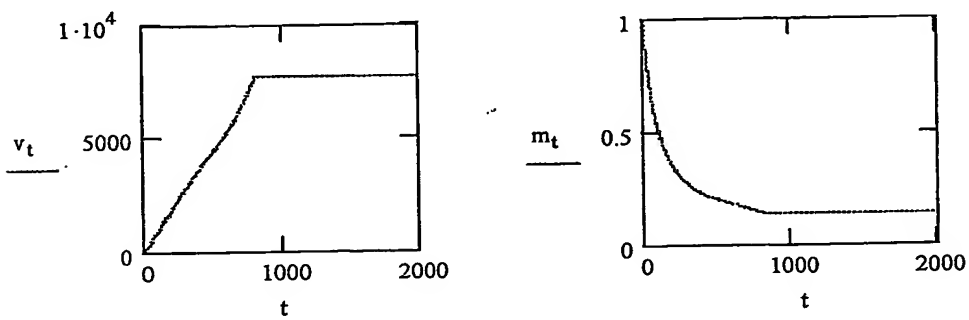
$$m' = -\frac{2E}{p^2}$$
$$u = \left(p^2 - m^2 g^2 \frac{\left(vorb^2 - v^2\right)^2}{vorb^4 m'^2}\right)^{1/2}$$
$$v' = \frac{2E}{p^2 m}\left(p^2 - m^2 g^2 \frac{\left(vorb^2 - v^2\right)^2}{vorb^4 m'^2}\right)^{1/2}$$

If $m(0) = 1$, E = 30,100 then v and m now progress as follows:



The ratio of initial to final mass is c. 4.67 and so the power per unit final mass is now c. 140 kW kg^{-1}.

C.8 Better therefore is for the propellant ejection speed to take the value calculated as in C.5 until the maximum propellant ejection speed is reached, and then to continue ejecting propellant at that speed, i.e. a combination of C.5 and C.7. If the maximum propellant ejection speed is 10,000 ms^{-1}, $m(0) = 1$ and $E = 10{,}000$ then v and m now progress as follows:



The ratio of initial to final mass is c. 7.1 and so the power per unit final mass is now c. 71 kW kg^{-1}.

C.9 In the majority of the cases considered above, the power per unit final mass is dependent on the choice of $E/m(0)$. I have chosen values of $E/m(0)$ that seem to produce relatively low power per unit final mass, without resulting in an excessively high ratio of initial to final mass.

C.10 Each 1 m^2 of collector area perpendicular to the sun's rays would collect circa 1.37 kW of solar power, given the strength of the sun's radiation in the vicinity of the earth. So in the theoretically optimal case given by C.5, each 1 kg of final mass would require circa 45 m^2 of collector aperture area perpendicular to the sun's rays to generate sufficient power to put the 1 kg into orbit (assuming 100% energy efficiency and no other weight elsewhere).

A practical mirror arrangement might require only circa 1.1 times this area of mirror, see Appendix A. Such a mirror would therefore weigh about 0.25 kg if the mirrors were made of material weighing 5 gm^{-2}. This rises to c. 0.30 kg if the propellant ejection speed is limited to 10,000 ms^{-1}.

C.11 There is probably a sufficient margin between 0.30 and 1 to make it possible to consider using this mirror technology in this fashion, although further reductions in mirror weight would considerably improve the practicality of a solar-powered orbital launch vehicle.

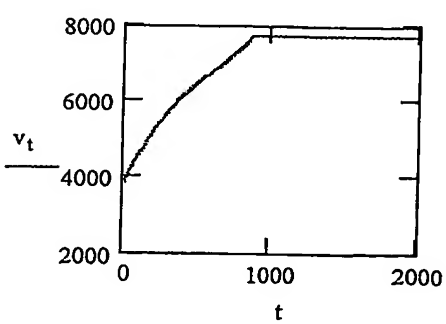
The highest specific impulses currently available (other than via solar sails) involve gridded ion engines and propellant ejection speeds of c. 40,000 ms^{-1}. The required power per unit final mass if a limit of 40 kms^{-1} is placed on ejection speed is about 63 kW kg^{-1}, i.e. very close to the

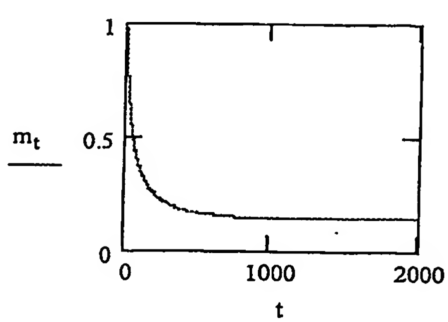
theoretically optimal identified in C.5. However, you would in practice need a two-stage process to convert the sunlight into electricity and then use the electricity to power the ion drive. The efficiency of conversion of sunlight into electricity seems unlikely to be much above 25-30%, and the efficiency of conversion of electric energy into thrust seems unlikely to be much above 60%, according to [2]. Thus the overall energy efficiency is unlikely to be better than c. 15%. This would increase the required mirror weight six-fold. Also, even using the lightest sunlight to electrical energy conversion approach (which is likely to be thermionic power conversion, see Appendix D) there are other components that could add significantly to the weight of the overall launcher, e.g. power regulators. This suggests that the mirror weight per unit area might need to fall a further 15-30 fold for such an approach to provide a practical means in isolation of launching vehicles from rest into earth orbit.

C.12 Instead, I think that a more hopeful propulsion technology in this respect is solar thermal propulsion. The likely maximum propellant ejection speed would be c. 10,000 ms^{-1} if the propellant were H_2 heated to circa 3700 K. This increases the required power per unit final mass to the c. 71 kW kg^{-1} identified in C.8. When operating at 3700 K approximately 15% of the incident sunlight falling onto such a solar thermal engine would be reradiated out (note: if the engine were at thermal equilibrium with the sunlight, i.e. were at the same temperature as the sun's photosphere, then this figure would rise to 100%). The solar thermal rocket engine should behave like a gas turbine heat engine, and if, say the propellant were ejected at a temperature of 1000 K then the thermodynamic efficiency of the engine is subject to an upper limit of around 70-75% (i.e. (3700-1000) divided by 3700). Therefore the net conversion efficiency of the energy in the sunlight to thrust is likely at best to be c. 60%, and more probably c. 40 – 50%. Such an engine should be simpler in concept and therefore probably lighter than the more complex thermionic power plus ion engine, thereby saving further weight. If these sorts of efficiencies are achievable then such a launch vehicle might still have power to spare to launch payload in addition to the mirrors into orbit.

C.13 More practical in the short term may be to consider a *chemical rocket assisted solar-powered* launch vehicle (particularly if the same engine chamber is used for both the chemical rocket and the solar thermal propulsion mechanism, since this could reduce the vehicle weight. It would seem beneficial from an energy budgeting perspective, if you are planning to use both chemical and solar powered rocketry in the same launch vehicle, for the solar powered rocketry to be used after the chemical rocketry in the launch sequence. This ordering also has the advantage that the chemical rockets could lift the vehicle above the (lower) atmosphere before deployment of the mirrors used to concentrate the solar power, overcoming the problem of high air resistance such mirrors would otherwise face near the ground.

Suppose, for example, that chemical rockets were used to reach one-half orbital speed, i.e. c. 3,850 ms^{-1}, so that $v(0)=3850$, If $m(0) = 1$, $E = 3{,}000$ then v and m now progress as follows (from the time that the solar powered engine takes over):





The ratio of initial to final mass is c. 6.9 and hence the power per unit final mass (for the solar energy stage) falls from c. 60-75 kW per kg to circa 21 kW per kg (even if there is an upper limit on the propulsion ejection speed of 10,000 ms^{-1}). This result is relatively insensitive to the upper limit on the propulsion ejection speed, i.e. does not change much if this upper limit is changed between 8,000 and 40,000 ms^{-1}.This leads to a corresponding fall in the mirror weight of perhaps 60 – 75% Even a relatively modest boost from a chemical rocket (or air-breathing engine) early on in flight would seem to put the power to weight requirements much more plausibly within the reach of currently available mirror technology!

The faster such a launcher reached before it switched to solar power, the less is the required power to weight ratio needed from the solar energy stage. For example, if the chemical rocket could impart to the vehicle a speed of 5000 ms^{-1} (still less than half the kinetic energy it requires to reach orbit) then the required power to weight ratio from the solar energy stage falls by nearly another 50%.

APPENDIX D

The relative merits of solar thermal propulsion and solar thermionic powered engines and versus other propulsion and energy generation technologies

The Specific Impulses of Different Propulsion Technologies

D.1 The specific impulse of an engine is defined as the (effective) exhaust velocity (i.e. speed at which the propellant is ejected), divided by the acceleration due to gravity at the earth's surface. We refer to the "effective" speed because there is some, normally small, dependency on the nozzle design as well as on the propellant ejection speed.

D.2 In a chemical rocket, the specific impulse generally increases as the combustion temperature increases and the molar mass of the exhaust products decreases. Liquid oxygen and liquid hydrogen are nearly ideal chemical rocket propellants, particularly if their combustion is used to heat additional hydrogen, because they burn energetically at high temperature (about 3200 K) and produce non-toxic products consisting of gaseous hydrogen and water vapour, with a small effective molar mass (about 11 kg/kmol). The vacuum specific impulse of a rocket using these as fuel is about 450 seconds.

D.3 With solar thermal propulsion, the propellant is no longer heated by combustion but by sunlight. Hydrogen would normally be used as propellant, to minimise molar mass and hence maximise the specific impulse. In the SOTV concept, the concentrated sunlight would heat up liquid hydrogen to c. 2400 K, achieving a specific impulse of 750 to 850 seconds. The SOTV envisaged only a low thrust (in part because of the small size of the solar concentrators being proposed for it).

D.4 Electrically powered rocket engines can deliver higher specific impulses. A Hall effect ion thruster accelerates ions along the axis of the thruster by crossed electric and magnetic fields. A plasma of electrons in the thrust chamber produces the electric field. A set of coils creates the magnetic field. Such engines have flown on several Russian spacecraft and can deliver specific impulses of circa 1800 seconds.

Even higher specific impulses can be delivered by gridded ion engines. These typically use xenon as propellant. The xenon atoms are introduced into the thruster chamber which is ringed by magnets. Electrons emitted by a cathode knock electrons from the xenon atoms to form positively charged xenon ions. The ions are accelerated by a pair of gridded electrodes, and are then neutralised by electrons emitted by a further component, to prevent a space charge from building up around the satellite. These sorts of engines can deliver specific impulses of 2000 to 4000 seconds. They have flown on several commercial satellites.

D.5 Higher specific impulses imply lower amounts of propellant are required to accelerate the vehicle by the same amount. Current commercially available electric propulsion engines use propellant approximately 4 to 10 times more efficiently than chemical propulsion and approximately 2 to 4 times more efficiently than solar thermal propulsion.

D.6 Solar sails can deliver effectively infinitely high specific impulses. They provide thrust via the momentum imparted by changing the direction of photons striking the sail. They require no propellant at all.

Solar sails have been proposed for many years, but only recently has the technology evolved sufficiently to make them practical. The Planetary Society is planning to launch a test solar sail, COSMOS 1, into orbit. Its sails are made of 5-micron thin alumizied reinforced Mylar, which presumably therefore weighs circa 5 to 10 gm m^{-2}. NASA's Jet Propulsion Laboratory (JPL) is

reported earlier in 2002 to have made a semi-rigid carbon fibre mesh weighing only 5 gm m^{-2} (by "semi-rigid" is meant a material able to keep its shape even in the presence of gravity).

Application of the proposed concentrators to these technologies

D.7 The proposed concentrators can be applied to, and can in some cases improve on, several of these technologies:

(a) **Solar thermal propulsion**

The proposed concentrator design can in principle deliver temperatures up to c. 5900 K outside the atmosphere. This is higher than the melting point of any element that could be used to enclose the heating chamber. The materials with the highest melting points appear to be tungsten (highest melting point of all metals, circa 3680 K) or carbon (melting point circa 3800 K). Operating at these latter temperatures, the specific impulse a solar thermal engine would provide, using hydrogen as propellant, would rise to circa 1000. The very small area onto which the sunlight is concentrated means that less of the incident energy is reradiated away even at these high temperatures, improving the overall energy efficiency of the engine. Assuming that the rocket engine otherwise behaves like a tolerably efficient heat engine with rejection temperature equal to the temperature at which the hydrogen might be expelled, say 1000 K, then heating the hydrogen to initially 3700 K should make possible an overall efficiency of conversion of the energy in the sunlight into propulsion of 40 - 50%, see Appendix C.11.

Like chemical rocket engines, conventional jet engines work by expelling gas that has been heated by a chemical reaction. The main conceptual difference is that they burn oxygen sourced from the air rather than stored within the vehicle. A way of increasing the thrust of a jet engine is to include "after-burn", i.e. to ignite more fuel downstream from the air intake and compressor. This increases the speed at which the propellant is ejected. Typically it is not energy efficient to do this, so afterburners are generally confined to military jets and used only when added thrust is particularly helpful.

It should in principle be possible to use the same sort of technique to boost the specific impulse available from a solar thermal engine. In a classical chemical rocket or solar thermal engine, the propellant is heated to a high temperature and pressure in the combustion chamber or equivalent. The gas then passes through a narrow rocket throat and subsequently expands as it passes through the nozzle, gaining ejection speed as it does so. The nozzle needs to allow the gas to be expelled without obstacle, as to do otherwise would reduce the ejection speed. However, if it were possible to heat the gas during this expansion phase without introducing obstacles to its free flow, then this would allow the gas to expand further increasing the ejection speed. This might be achievable if the nozzle were itself long and at a high temperature and so radiated energy into the expanding hydrogen gas. This might perhaps be facilitated by embedding most of the nozzle, perhaps in a spiral shape, within the slab onto which the sunlight was being concentrated. The introduction of trace heat absorbent substances into the heated hydrogen gas might improve the radiative heat transfer from the nozzle sides to the gas within.

The amount of heat radiated by a black body increases as the 4th power of the temperature. Thus increasing the temperature to which the slab is heated from, say, 2400 K to 3700 K would increase the radiative heat transfer more than five-fold, making a material improvement in the specific impulse more likely with the proposed concentrator design than with other alternatives.

Whether the resulting potential improvement in specific impulse would be sufficient to justify the probable reduction in energy efficiency is unclear to me, and the possibility of such "after-throat assisted" solar thermal propulsion is not considered further in this paper.

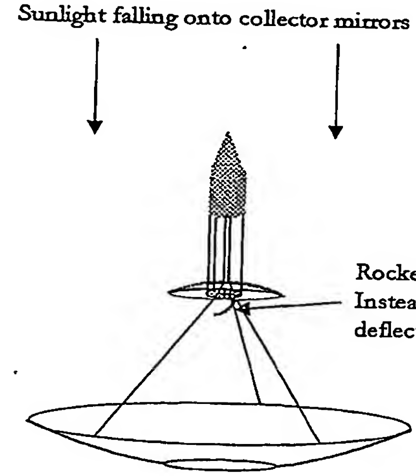
(b) **Solar-powered electric engines**

The most obvious way in which the proposed concentrator design could be used to improve the efficiency of existing solar-powered electric engines is by improving the efficiency of the power generation. For example, the high temperature such a concentrator can deliver might make solar thermionic power more practical, see below.

(c) **Solar sails**

If the proposed concentrator were light enough to lift its own weight in the pre-orbital trajectory phase, then it should also be light enough to function as a solar sail, without much extra effort. The sunlight could be concentrated onto a parabolic mirror near the image point attached to the payload, with the positioning of this additional the parabolic mirror would dictate the direction of thrust. But this parabolic mirror would be small in relation to the rest of the mirror arrangement, and therefore add little to the overall weight of the vehicle. The proposed mirror layout would have the added advantage of having a smaller overall surface area than more traditional solar sail designs arrangements, which means that it would probably in aggregate weigh less.

Figure 4



However, it is worth noting that solar sails do have the disadvantage that their specific impulse is so high that they generate very little acceleration. The momentum of a photon is its energy divided by the speed of light. So suppose a vehicle of the sort proposed earlier in this paper used mirrors that weighed 5 gm^{-2} and suppose that it had lifted itself plus its own mass into orbit. If the mirror layout was then used as a solar sail at a distance from the sun similar to that of the earth then it would generate an acceleration of $(1370/(2\times0.005)/(3\times10^8) = 4\times10^{-4}$ ms^{-2}. A journey of, say, 10^{11} m (approximately 2/3rds of the distance from the earth to the sun), would take 260 days, if you were accelerating from rest at this rate and with this acceleration, with the spaceship ending up at a speed of circa 9,000 ms^{-1} by the end of the trip, i.e. not much more than the speed the ship had reached in its pre-orbital trajectory.

Better may be to devise more effective ultra-high specific impulse solar-powered electric engines. For example, suppose one could create a "balanced" ion engine that ejected equal amounts of positive and negative ions. The ejected ions should then be attracted to each other rather than back towards the spaceship, which is the usual difficulty that an ion engine faces. Particularly attractive ions in this context might be H_2^+ and H_2^- since H_2 is the lightest easily available gas, and these ions can thus be accelerated to a particularly fast speed per unit of applied electric field. The mass of a proton is 938 MeV, so the speed that either of these ions can reach if accelerated by a 1 MV electric field is circa 3% of the speed of light, i.e. circa 10^7 ms^{-1}, or if by a 10 kV electric field to

circa 10^6 ms^{-1}. The precise method used to accelerate the ions to this speed essentially involves taking existing particle accelerator technology and making it as lightweight as possible. Suppose you could arrange for ejection of these ions at 10^6 ms^{-1}, and you could achieve a 10% overall energy efficiency for conversion of the sunlight into thrust. Suppose also that the overall spacecraft mass was, as before, twice that of the mirrors, and the spacecraft was a similar distance from the sun as the earth. Then this sort of engine would create an acceleration of c. 3×10^{-2} ms^{-2}. In 20 days it would have accelerated to circa 25,000 ms^{-1}, having ejected propellant with a mass of only circa 5% of the mass of the mirrors. Solar sails would probably ultimately only be competitive against such an approach in instances where sedate speeds were acceptable.

Space-Based thermionic power generation

D.8 The US Government's DTRA has been carrying out some research into thermionic power generation, including thermionic power generation for space missions. The US National Research Council Committee on Thermionic Research and Technology recently reviewed this research, see [2].

D.9 Thermionic energy conversion is a process that converts heat directly into electric power. In its most elementary form, a thermionic converter consists of two metal electrodes separated by a narrow gap. One of the electrodes, called the emitter, is held at a high temperature, perhaps c. 2000 K. The other electrode, called the collector, is held at a lower temperature, perhaps c. 1000 K. The emitter emits electrons into the gap and the lower temperature collector absorbs them. The binding energies of the emitter and collector surfaces that act on the electrons are known as the work functions of the electrode surfaces. The electrons absorbed by the collector produce a usable electrical current as the return to the emitter through an external circuit. Electrical power is produced by virtue of the potential difference between the emitter and the collector.

A thermionic power generator can be thought of as an idealised heat engine that uses electrons as its circulating fluid. It is a static device that has no moving mechanical parts. It operates at high temperatures, generates high power, and occupies only a small volume. For these reasons, thermionic converters have been considered potentially useful as power sources for use in space as well as for high temperature terrestrial power systems. Research into thermionic power generation appears currently to be proceeding in Russia, Sweden and China.

The high heat rejection temperature, typically 1000 K or more, allows thermionic systems to use compact radiators with relatively low mass. This appears to offer weight advantages over most other power conversion candidates, the notable exception being liquid metal rankine cycle converters.

D.10 Efficient operation requires and emitter surface with a relatively low work function of 2 electron volts or less and an even lower collector work function of 1.5 electron volts or less. The material most often suggested in this context is tungsten, aided by cesium vapour in the gap between the emitter and collector.

D.11 Thermionic power generation only becomes practical at very high temperatures, but these are readily available via the mirror layout described above. Indeed, care might be needed as otherwise the temperature reached could exceed even the melting point of tungsten or any other material. As the above mirror layout is aplanatic, the surface area of the power generator could be about the same size as the area onto which the sunlight is being focused (i.e. only c. 1/40,000th of the collector aperture area, this fraction being about one divided by the square of the angle subtended by the sun at the earth's surface).

D.12 Solar thermionic power generation was explored briefly by NASA in the 1960's. This included validation of the concept by NASA under the Solar Energy Technology (SET) program, involving

converters operating at 25 Watts per square centimetre and 0.7 volts, over 15,000 hours and through several hundred thermal eclipse cycles.

The research was curtailed in the early 1970's in favour of solar photovoltaic battery systems. Photovoltaic systems with battery storage are the most commonly flown space power systems. They have been in use since the late 1950's and have flow on more than a thousand missions. Photovoltaic systems have been designed to supply up to 75 kilowatts of electricity for up to 15 years of life.

D.13 The greater the power requirements of a space vehicle, the more likely is solar thermionic power generation to prove attractive versus other power generation approaches, particularly solar photovoltaic power generation, in terms of stowed payload volume and mass. One of the DTRA's contractors, General Atomics, proposed a high-power, advanced, low mass (HPALM) solar thermionic converter. The concept involves the use of an inflatable solar concentrator to focus solar energy onto a thermionic converter to supply power to a spacecraft. It uses a large off-axis, inflatable parabolic reflector to focus solar energy into a 2000 K heat receiver that is radiatively coupled to the thermionic devices. Heat pipes would be used to remove the waste heat, at approximately 1100 K, from the collector surface. General Atomics anticipated reaching efficiencies of 20 to 25 percent for the converters. This is a pretty impressive efficiency, given that the theoretical upper limit on the conversion efficiency set by the Second Law of Thermodynamics with such input and output temperatures is only (2000-1100)/2000 = 45%. The HPALM concept was sized to provide 50 kilowatts of electricity and with a calculated performance of 106 watts per kilogram and 80 kilowatts per cubic meter on a stowed configuration ready for space launch.

Such a system would, General Atomics claim, take up less room on the launch vehicle and weigh about the same as a solar array of current design for a 20 kilowatt communications satellite. Below 20 kilowatts, however, solar thermionics is less likely to prove competitive.

D.14 The Committee noted that currently the approximate cost to launch a spacecraft is $6,000 to $10,000 per kilogram to reach low earth orbit (LEO) and $20,000 to $40,000 per kilogram to reach a geosyncronous earth orbit (GEO) depending on launch vehicle type. Basing its analysis on the lower value of $20,000 per kilogram to GEO, the Committee concluded that a 100kW solar thermionic power system might offer the following financial benefits:

The cost of a standard (chemical) rocket capable of the sort typically used to move a vehicle from LEO to GEO is c. $3.5m. The Committee estimated that the cost of an electric propulsion system plus a 100 kW space power generation system would be approximately $9m + $33m = $42m (ignoring amortisation of the costs of developing a working solar thermionic power generator). Depending on how the power was used, there are two situations.

(a) If the 100kW power source is used purely for electric propulsion only, the approximately 3,0000 kilogram savings in mass equates to a saving of $60m less extra costs of $39m generating a total net saving of $21m

(b) If the 100kW power source is used for both propulsion and primary mission power once on station, the approximately 6,000 kilogram savings in mass yield an approximately $120 million worth of additional payload for an approximate $114m net savings.

The savings available from dual use, i.e. (b), are substantially greater than for single use, i.e. (a).

D.15 Overall, the Committee concluded that a space solar power system was the most promising near-term application for thermionic technology and it recommended that the US Government should concentrate its near-term thermionic development work on a system such as the high-power, advanced, low mass (HPALM) concept or the possible use of thermionics in the context of the proposed solar orbital transfer vehicle (SOTV).

However, while space-based thermionic power generation was the most promising application of thermionics, the Committee also cautioned that success was not certain. The history of spacecraft performance demonstrates that it is difficult to compete with photovoltaic systems. Significant progress continues to be made in photovoltaic converters. For instance, triple junction solar cells can now deliver up to 29% efficiency.

The Committee also noted several specific issues that need to be addressed before space-based solar thermionic power might become a reality, including:

(a) Power conditioning. Thermionic power converters generally work on low voltages, e.g. 1 V or less, and high currents, whilst working at high temperatures. Unless you had large numbers in series, they would require power conditioning systems to raise the voltage to the higher levels 100 to 300 V that current or proposed spacecraft might use. A recent power conditioning uing developed for a 5 kilowatt Hall effect propulsion system that converted 28 volts to 300 volts weighed approximately 25 kilograms.

(b) Energy storage. Energy storage for satellites in planetary orbit could be via batteries, as per current photovoltaic systems, or via thermal storage systems, but the former reduce the potential weight benefit of solar thermionic converers and the latter are untested.

Terrestrial Solar Thermionic Power Generation

D.16 The Committee noted that terrestrial thermionic power applications have received little attention from any research organisation over the past two decades. The Committee believes that this lack of interest is a result of the high (development) cost of thermionic systems and the fact that neither long term reliability nor the systems themselves have yet been proven.

In the early 1960's the American Gas Association and the US Army started funding programs to develop fossil fuel powered thermionic converters. The material silicon carbide emerged as the preferred coating for emitters to protect them from air and combustion products. Because emitter materials, such as tungsten, had thermal expansion coefficients different from that of silicon carbide, cracking and separation problems were severe. These problems were not overcome until the 1980's by which time there was no funding available to demonstrate the practicality of fossil fuel powered thermionic devices. Thermionic topping cycles have some potential benefits, but have not been commercialised due to high capital and operating costs as well as the expense of developing initial prototypes. When the Committee wrote their report, they were doubtful whether any viable terrestrial applications then existed for commercial use of thermionic power conversion.

D.17 There have been some recent advances that may make either space-based or terrestrial based thermionic power generation more attractive. These include using oxygenated niobium to reduce the work function of the collector, and the development of microminature thermionic converters.

The Solar Orbital Transfer Vehicle Program

D.18 The Solar Orbital Transfer Vehicle (SOTV) concept vehicle program carried out the US Air Force AFRL proposed a parabolic mirror arrangement and a combined receiver/absorber/converter device that generated propulsion using solar thermal propulsion and then generated electric power for onboard vehicle use via solar thermionic power generation. The SOTV is a follow-on to the hydrogen fuelled Integrated Solar Upper Stage (ISUS) Orbital vehicle program started in 1994. During this program, some String Thermionic Assembly Research Testbed (START) tests were initiated. The logic was that as the ISUS concept vehicle would need very high temperatures for the hydrogen fuel propulsion, thermionics might be an appropriate power conversion technology

since it also required a high temperature heat source. The START tests seem to have gone worse than expected, indeed were worse that equivalent tests that JPL carried out in the 1960's under the JET programme. The Committee therefore concluded that these tests were unrepresentative of what solar thermionic power generation might actually deliver in the future.

The optimal propulsion technologies that a vehicle should adopt at different stages in its journey (if it is to use solar energy as an engine power source for some of its journey)

D.19 Appendix C highlights the increase (in theory without limit) in the specific impulse that is optimal for a solar powered launch vehicle to deliver as it approaches orbital velocity. After orbit is reached, gravity may be largely ignored, so very high specific impulses are always to be preferred (unless there is some benefit in arriving faster – which will almost always be the case if only because of the time value of money).

Bearing in mind the likely energy efficiencies and specific impulses deliverable from different propulsion technologies, there seems to be a progression in the technologies that are optimal at different stages in a journey that an interplanetary spacecraft might make.

In the absence of further improvements in ultra-lightweight mirror technology, this progression seems involve the following (in the order quoted): chemical rocketry, solar thermal propulsion, solar electric propulsion and finally, if speed is not of the essence, solar sail propulsion (at least as long as the vehicle is not travelling very far away from the sun).

A vehicle design like the one being proposed, that permits this optimal progression to occur and reuses vehicle components when doing so, seems likely to offer attractive payload and hence cost advantages.

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